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CONTRACT REQUIREMENTS	CONTRACT ITEM	MODEL	CONTRACT NO.	DATE
Exhibit E	5.1	LEM	NAS 9-1100	14 Jan 63

Type II Documentation

Primary Code No. 013

REPORT

NO. LED-510-1 DATE: 3 April 63

PRELIMINARY

"FIRE-IN-THE-HOLE"

STUDY [U]

CODE 26512

~~AVAILABLE TO NASA HEADQUARTERS ONLY~~

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REVISIONS

DATE	REV. BY	REVISIONS & ADDED PAGES	REMARKS
		CLASSIFICATION CHANGE	
		To UNCLASSIFIED	
		By authority of ~~ab Folliotte~~	
		Changed by L. Shirley Date 12/31/72	
		Classified Document Master Control Station, NASA	
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SUMMARY

This report represents a preliminary evaluation of the "Fire-in-the-Hole" problem connected with ascent engine light-off. Since the complicated flow field around the descent engine makes a completely analytical study difficult, the major portion of the effort to date has been devoted to construction and test of a 1/10 scale cold flow model. Over 30 test runs have been completed. These tests have demonstrated that improper venting arrangements can result in several different types of failure of the LEM vehicle. However, the present descent stage design is adaptable to a wide range of venting configurations without major weight penalties. Initial test results indicate that a 1000 in^2 minimum annulus around the full scale descent engine plus a stage separation of two inches would provide sufficient vent area. Subsequent phases of this program are expected to improve the accuracy of these predictions.

DISCUSSION

The "Fire-in-the-Hole" problem refers to the flow of the ascent engine exhaust over the descent engine at ignition of the ascent engine. Based on the ascent engine developing rated thrust and with no additional pressure force, the ascent stage will have lifted five feet off the descent stage after the first second. During this time several types of failures are possible if the ascent exhaust gases are not properly vented. If the vent area is insufficient, high pressures will be built up in the descent stage. This could conceivably result in structural failure in the descent stage. More important, however, is the effect on ascent engine operation. As the engine is back-pressured, flow separation and possible shocking will occur in the ascent skirt. This may result in pressures and heating rates in excess of reasonable skirt design capabilities. If the back-pressure is high enough, the ascent nozzle will become unchoked resulting in excessive combustion chamber pressure and probable failure. In addition to providing sufficient venting area to prevent ascent engine back-pressure, the vent area must be arranged so that any unsymmetrical pressure forces on the ascent stage are within the corrective capability of the reaction control system. Another problem to be considered is heating caused by the ascent exhaust. Although the time period during which this can occur is short, the lines and tanks of the descent stage must be configured to insure that no fuel is ignited by overheating from the ascent engine plume.

The area thru which the ascent exhaust gases flow contains the descent engine, support structure, gimbal actuators, valve packages and associated electronics and three fuel and three oxidizer lines. A model test was considered essential to determine the flow field under these conditions.

A model test program began March 8 at Grumman using a 1/10 scale cold flow model. Nitrogen gas ($\gamma = 1.4$) is used. In order to simulate the full scale base pressure relationships for the combustion products ($\gamma = 1.2$) the following scaling laws were used. (references 1 and 2).

$$\left(\frac{\gamma M^2}{\sqrt{M^2 - 1}}\right)_{Model} = \left(\frac{\gamma M^2}{\sqrt{M^2 - 1}}\right)_{Full\ Scale}$$

M = Ascent Engine Exit Mach. No.

γ = Ratio of Specific Heats

$$\frac{P_{c\ Model}}{P_{c\ Full\ Scale}} = \frac{\left[1 + \frac{\gamma - 1}{2} M^2\right]^{\frac{\gamma}{\gamma-1}}_{Model}}{\left[1 + \frac{\gamma - 1}{2} M^2\right]^{\frac{\gamma}{\gamma-1}}_{Full\ Scale}}$$

P_c = Ascent Engine Chamber Pressure

The resulting relations between the model and the full scale ascent engine are as follows:

Full Scale	Model
Chamber Pressure $P_c = 120$ psia	$P_c = 31$ psia
Area Ratio $\frac{A_E}{A_T} = 40$	$A_E/A_T = 7$
Exit Mach. No. $M = 4$	$M = 3.53$
Exit $\gamma = 1.2$	Exit $\gamma = 1.4$

Using these parameters the model static pressures closely simulate the full scale pressures in the descent stage and the model thrust scales as the area (.01 Full Scale).

An analytical study is progressing concurrently with the model test program. When the analytical flow field for $\gamma = 1.4$ has been verified by the cold flow tests the analysis will be applied to the full scale ($\gamma = 1.2$). If deemed necessary from this analysis a hot flow test series will be initiated.

The test setup consists of two 125 ft^3 supply tanks feeding thru 2" lines with solenoid-operated globe valves to a 4" Y section. The 4" section converges to a 2" section which represents the ascent combustion chamber. The two globe valves have a 65 millisecond response which very closely simulates the actual combustion chamber pressure buildup. The model is mounted in a small chamber with the thrust vector horizontal. The small chamber exhausts thru a 10" port into a 10,000 ft^3 high vacuum chamber. A picture of the test setup is shown in Figure 1. A view looking into the descent engine mounted in the chamber is shown in Figure 2. Tests are run for approximately one second at an initial chamber pressure of 10 microns. Final pressure is about 2000 microns. This final pressure is sufficiently low so that the flow in the descent engine is not noticably affected. The instrumentation consists of GAEC strain gage force pickups to measure the force between the stages and Statham differential and absolute pressure transducers. The readout equipment consists of CEC 7-315 galvanometers, an 18 channel CEC direct writing oscillograph and an 18 channel GAEC strain gage balancing unit.

The model used in the initial tests consists of ascent and descent stages each with a circular area 14.6 inches in diameter as the separation surface. These surfaces are connected by three strain gage force links located 120° apart. The descent engine is mounted in slots to the hexagonal inner surfaces of the descent stage. The engine is movable forward and aft to vary the annular vent area. The stage separation can also be varied. The hexagonal configuration was the existing design at the time the model was constructed and has the additional advantage of being adaptable to Schlieren photography. The pressure data will be correlated with the flow field as determined from Schlierens. A picture of the model is shown in Figure 3.

The "Fire-in-the-Hole" test program has been divided into several phases as follows:

Phase 1 - March 8-22 - Preliminary evaluation of test setup, and limited acquisition of data. The test setup has been verified, the back-pressure of the small chamber has been found to be within allowable limits, the valve response simulates the actual combustion chamber, the scaling relationships appear adequate and the instrumentation has proved satisfactory. Data has been taken on the force between ascent and descent stages and readings from 6 pressure taps have been recorded.

Phase 2 - March 25-May 6 - Engineering data acquisition. Extensive pressure and force measurements are being made for a wide variety of porting arrangements. The model currently is instrumented with 15 pressure transducers and three strain gages. A diagram of the model showing the current instrumentation is shown in Figure 4. High speed Schlieren movies will be taken of the flow field thru 6" viewing ports. The glass model is being constructed and movies should be taken beginning April 15.

Phase 3 - (if required) - May 6 - Additional engineering data. Increase Schlieren to 12" field to cover entire descent stage. Run actual separation tests of an ascent stage.

Phase 4 - (if required) - Hot Flow Tests

The data to be taken during Phase 2 will consist of evaluating the effect of varying the x, y, z dimensions shown in Figure 4.

The data taken so far has been for Y = 0 and Y = .25 and X and Z varying by changing the position of the descent engine. The effect of varying X is small as long as the normal shock over the descent engine does not back up into the ascent engine. For all tests run so far the shock standoff was well below the ascent engine exit plane so the main effect on the flow field when the engine is moved, is due to varying the exit annulus (Z dimension).

As this area is opened up the flow around the descent engine should change from subsonic with a choked exit to supersonic with a choked exit to completely supersonic flow. This area was varied from 10 in^2 to 15 in^2 with $Y = 0$. The flow, for the range of annular vent areas tested, is supersonic with the exit sonic or low supersonic. The most important parameter, the base pressure on the ascent stage, varied from .7 to 1.1 psia. Even at the minimum vent area tested (10 in^2) this back-pressure is barely sufficient to cause separation in the ascent engine. For the full scale engine with a design pressure ratio (chamber to exit) of 456 it is estimated that flow separation will occur at a pressure ratio (chamber to ambient) less than 123. For the model the design pressure ratio is 84 and separation will occur below a pressure ratio of 29. (Reference 3). At the minimum area tested the pressure ratio was approximately 29. It can therefore be tentatively concluded that an annular vent area of 10 in^2 (1000 in^2 full scale) will be sufficient to prevent separation especially when an additional vent area (Y dimension) is provided between the stages.

The provision of vent area between the stages is in fact a requirement. This results from measurements of force between the stages. On the initial runs the forces on the 3 strain gages which were located 120° apart, were very unsymmetrical. Although the stage separation (Y) was nominally zero the leakage thru the finite space between the stages produced a highly unsymmetrical force which was twice the thrust of the ascent engine. The moment produced by this force would far exceed the capability of the reaction control system to correct. In subsequent runs the stages were separated .25" (2.5" full scale). This known separation resulted in a much more symmetrical pressure distribution and a lower pressure level (due to the greater total vent area) but the total pressure force was still high, being approximately equal to the ascent thrust. The base of the model had a circular area equal to the area of the descent stage. The full scale ascent stage has a smaller area than the descent stage resulting in a lower pressure force than obtained from scaling the model results. However, the center of pressure of the full scale ascent base area, as presently configured, does not coincide with the ascent stage c.g. Dynamics studies are currently in progress to determine the maximum allowable moment due to the pressures between the stages. This requirement will be met by increasing the vent area to decrease the interstage pressure or by decreasing the moment arm by adding structure to the base of the ascent stage to position the center of pressure closer to the c.g. Model testing of the interstage pressures is not applicable to small spacing ($<$ .25") due to boundary layer buildup. ($\approx$ .030"). Pressures for the smaller spacing will be determined analytically.

For all tests the stagnation pressure on the top of the descent stage was lower than expected. A stagnation pressure survey of the ascent nozzle showed that the nozzle contour resulted in a large Mach number gradient. Instead of the nominal 3.53, the Mach number was approximately 4.7 at the center and decreased to 3.2 at the wall. The nozzle contour will not be changed until details of the exit flow field of the actual engine are received from Bell.

CONCLUSION

Our initial conclusions can be summarized as follows:

The range of vent areas considered appears sufficient to prevent back-pressuring of the ascent engine. These vent areas were consistent with the current descent stage design.

The pressure buildup in any finite space between the stages can significantly increase the force on the ascent stage. This force must be taken into account in determining vehicle dynamics during lift-off. This force can be decreased by opening up either (or both) the Z and Y dimensions to lower the pressure in the descent stage. The moment on the ascent stage can be reduced by adding the necessary structure to the base of the ascent stage to obtain the desired center of pressure.

The venting requirements incorporated in the present configuration studies are a minimum annulus of 1000 in^2 around the descent engine and a stage separation of two inches.

REFERENCES

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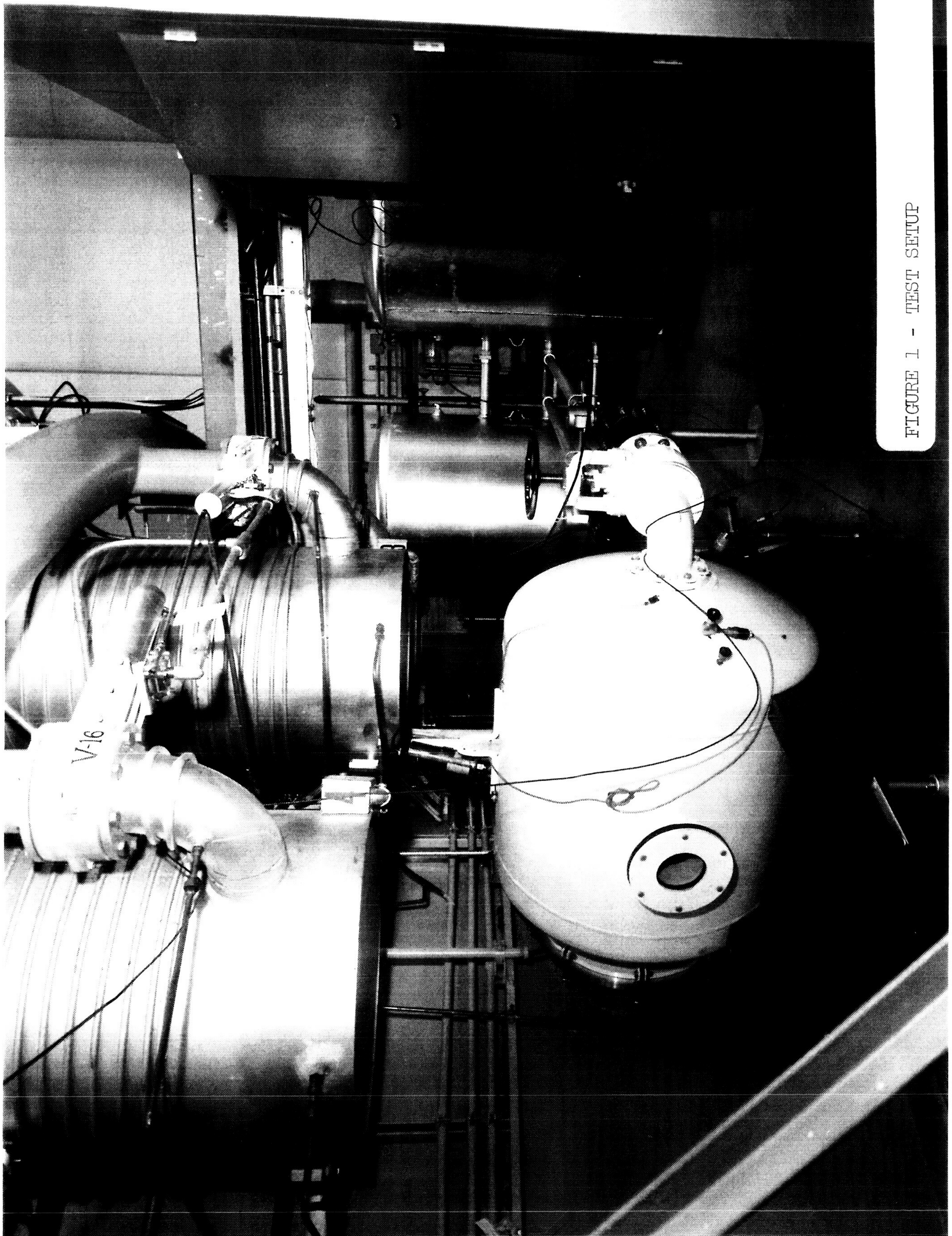

FIGURE 1 - TEST SETUP

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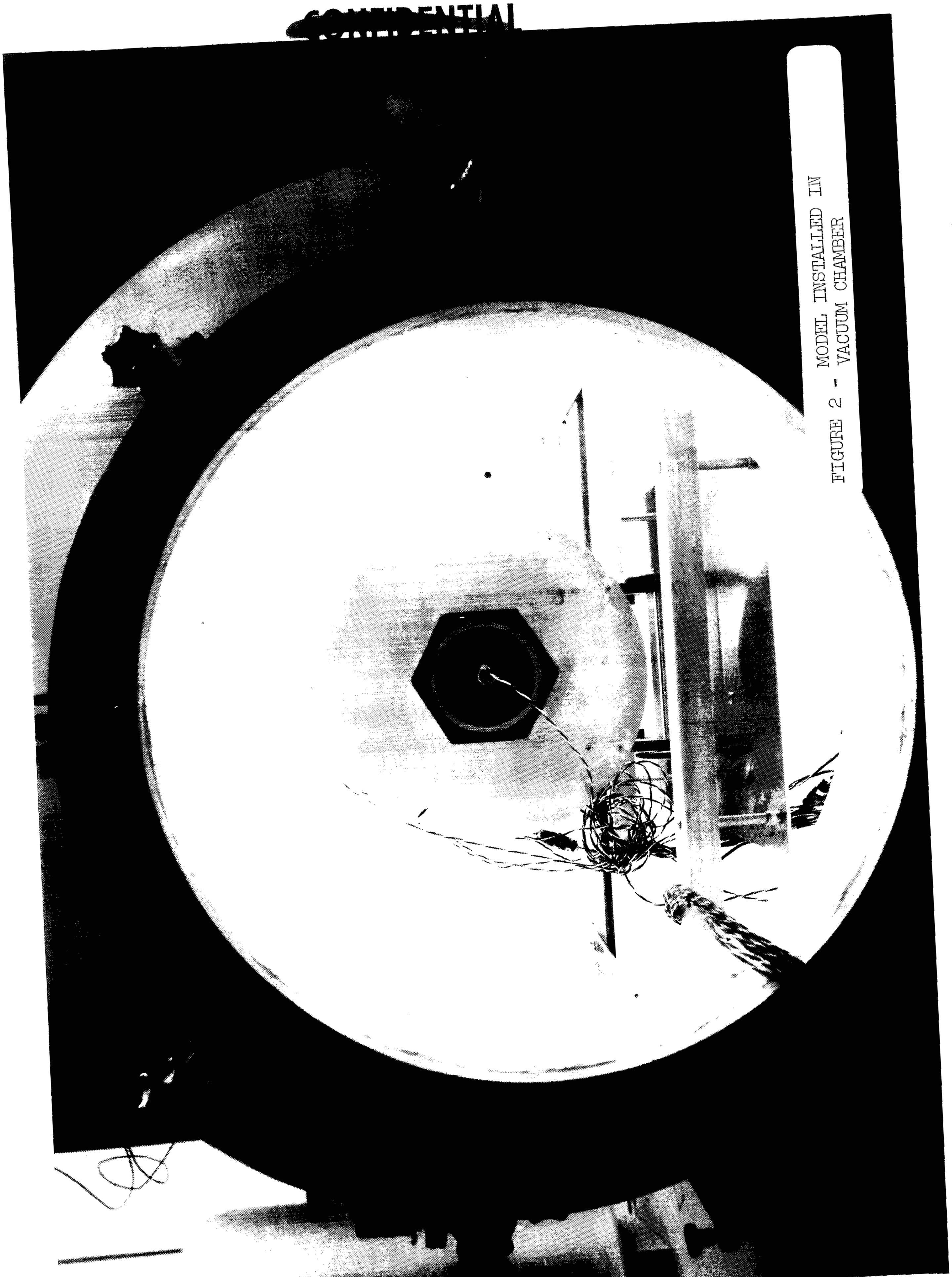

FIGURE 2 - MODEL INSTALLED IN VACUUM CHAMBER

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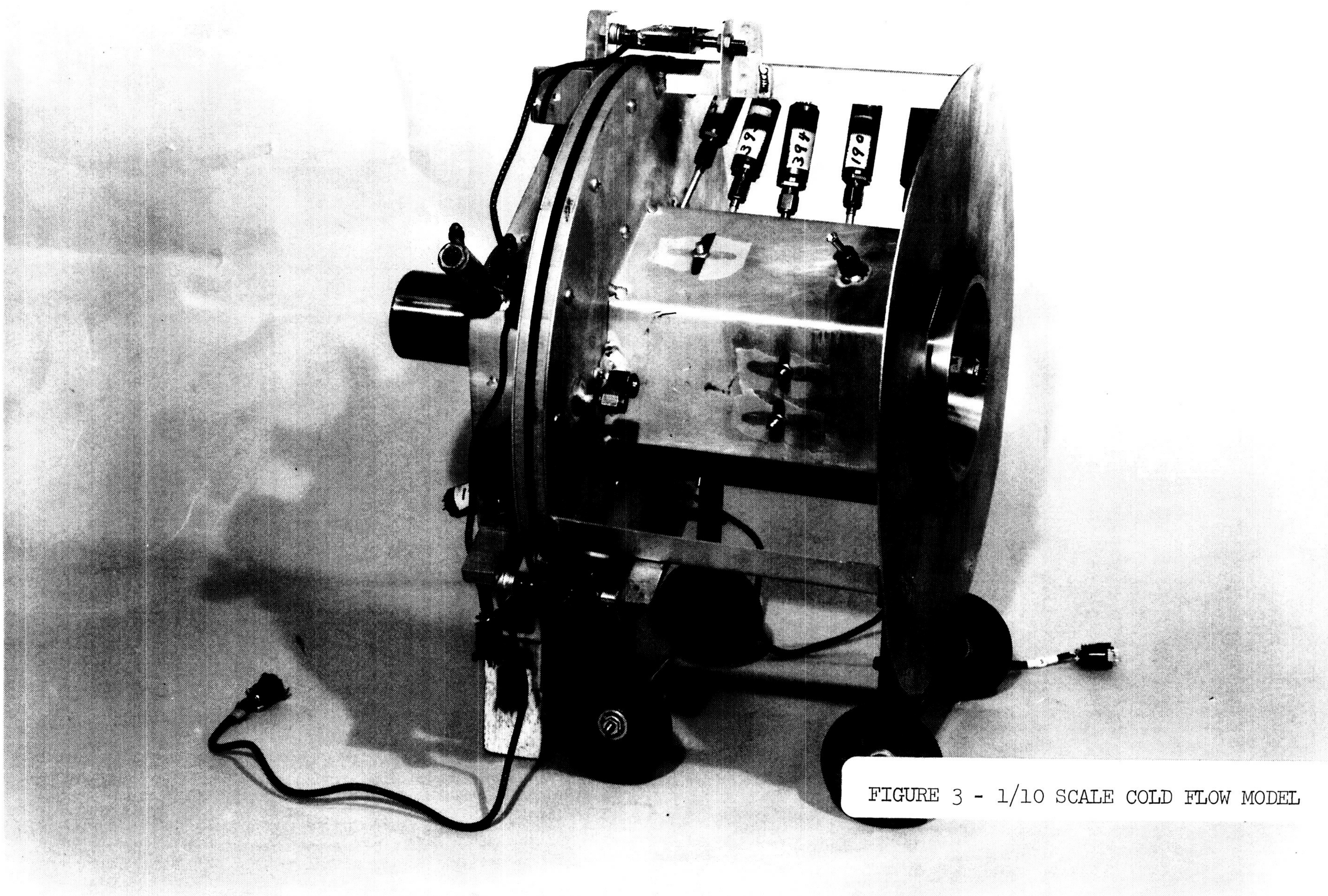

FIGURE 3 - 1/10 SCALE COLD FLOW MODEL

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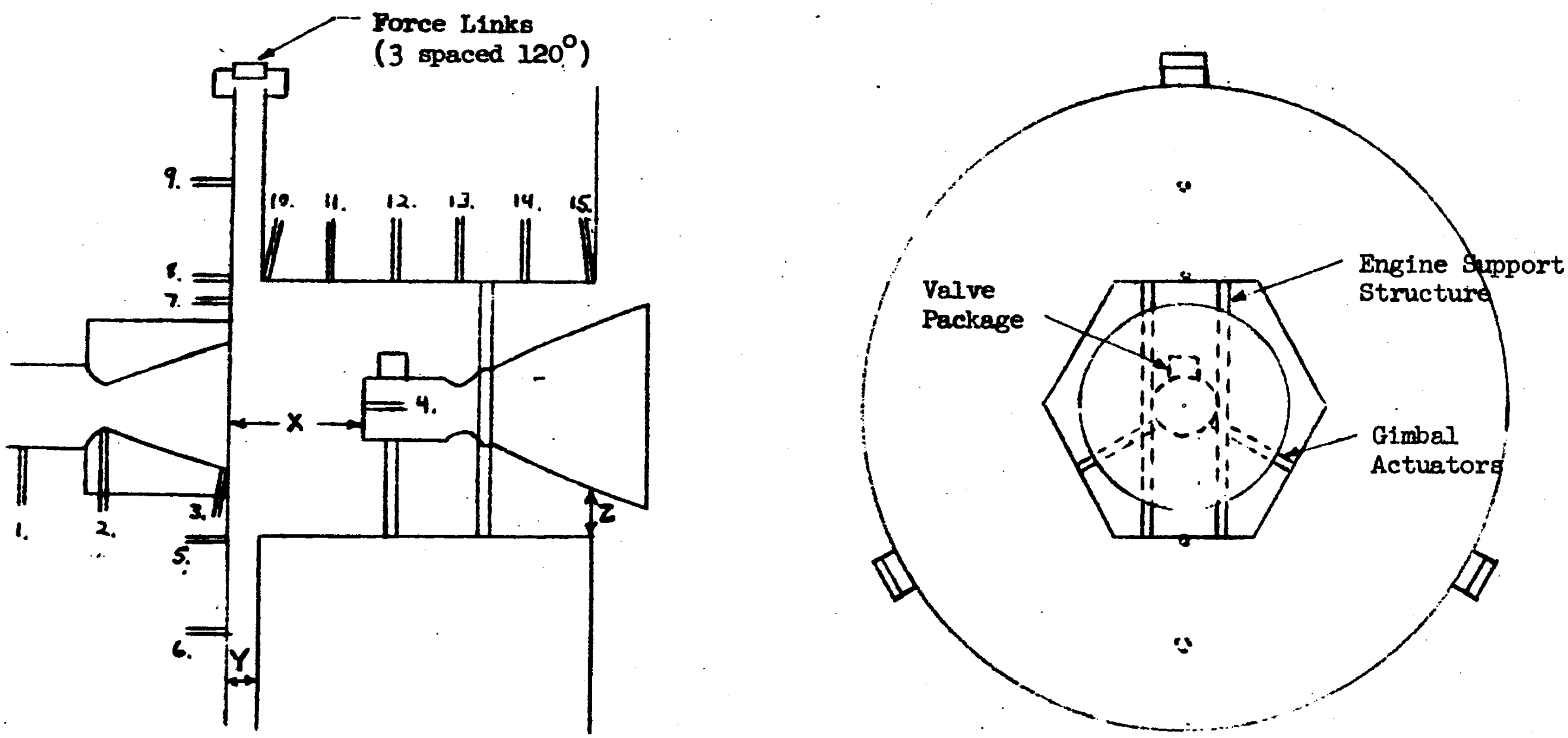

FIGURE 4

1/10 SCALE COLD FLOW MODEL

INSTRUMENTATION DRAWING

Current Instrumentation - 3 Force Links &
15 Pressure Transducers